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INVESTIGATION POTENTIAL FLOW ABOUT CURVED WING USING PANEL METHOD

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ABSTRACT

In the present investigation Low order panel method with Dirichlet boundary condition conjugated with Kutta condition, was used to calculate pressure coefficients for potential flow about isolated curved wings at different aspect ratios and different angles of attack. Also both local and total lift coefficients were calculated for the same wings, with detailed streamline behavior on both upper and lower surface. Constant strength quadrilateral doublet element and Constant strength quadrilateral source element were placed on each panel, except on wake sheet only constant strength quadrilateral doublets were placed to satisfy Kutta condition at trailing edge. A set of linear algebraic equations were established by setting inner potential equals to free stream potential. These equations were solved using Gauss-elimination to determine quadrilateral doublet singularity strength distribution. Finally finite difference formula was used to predict aerodynamic loads calculation.

KEYWORDS: Panel Method, Potential Flow, Curved Wing, Lift Coefficients

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Original Article

Nomenclature

AOAT	**Angle of Attack**	**Degree**		**□Φ***	**Total Velocity Potential Function**	**m²/s**
ASR	Aspect ratio	---		Φ	Velocitypotentialfunction	m^2/s
SBA	Swept back angle	Degree		□Φ_∞	AA Free stream velocity potential function	m^2/s
C_l	Left coefficient			□□	Vorticity strength	m/s
C_p	Pressure coefficient			□	, ma Angle of attack	Degree
n	Normal unit vector			□	Density	Kg/m^3
S_B	Solid body boundary			□	Source strength	m/s
Q□	Velocity vector	m/s		□□	Doublet strength	m/s
Q_∞	Free stream velocity	m/s		□		

1. INTRODUCTION

Aerodynamic models that can be used to compute the forces and moments on a wing orman aircraft configuration are useful in applications such as flight simulation, configuration design,flight-dynamics characterization, and control-system design[1].The enormous increase in air traffic in the last decades combined with increased public concern for the environment have led to an increased interest in reducing the environmental impact of aircraft[2].Aerodynamic analysis is the driving force in the preliminary design stage. Thechoice between theoretical, numerical or experimental aerodynamic analysis is relatively easy:numerical analysis in the form of computational fluid dynamics (CFD) provides not only thepossibility of computing flow simulations that are

either impractical or impossible to obtain inwind tunnels (high Mach, high Reynolds wind tunnels are still rare) but also the possibility oflowering the time and cost required for the range of simulations needed for the design of newaircraft [3].High fuel costs and environmental concerns provide continuing motivation for research aimed at increasingaircraft efficiency. Improved wing tip design can benet new designs and can be used toimprove performance by retrofitting existing wings[4].Panel methods are well known and widely accepted inaerospace and in naval industry for calculatingpotential flows[5]. It is not appropriate in modern aircraft design to ignore the influences of aspect ratio and sweep back angle in wings. Therefore, analysis for pressure distribution is very important for which considering effect. Accurate predictions of static aero elastic characteristics, which are especially important in the early design stage of aircraft, can be used to avoid revising the structural design repeatedly in the late phase of aircraft design[6].Many efforts have been made to develop accurate and suitable methods to investigate this problem. Panel methods have been widely accepted as a useful tool for aerodynamic and hydrodynamic design, since the pioneering work of Hess and Smith [7]. A large number of different panel methods have been developed for a variety of applications, leading recently to application to the analysis of steady performance[8].

NACA 0012 was used in our study to investigates pressure coefficient distribution, local lift coefficient distribution, velocity to free stream velocity ratio distribution on both upper and lower surfaces and lift coefficient for wing with difference sweep back angles (0,20,30,40,50 and 60) and different aspect ratios (4,6,8 and 10). For each case different angle of attack was studied to draw total lift coefficient behavior within range of (0- 12) degree.

2. CURVED WING CONFIGURATION

It’s important in aerodynamic industry to investigate new configuration, that may be good alternative for convential configurations. A new configurations may introduce a good improvement in aerodynamic performance, but at the same time develop a new technical challenges. for example the curved trailing edge in suggested wing may makes it more difficult to deal with flaps arrangement and controls at the same time choosing new designs may help hiding the flying object from radars, also it may improve aerodynamic performance in some cases. The suggested wing geometry is illustrated in figure 1 below.

For leading edge:

$$x = \frac{1}{2} span\ \tan(SBA)\ [\ 1 - (\frac{\bar{y}}{\frac{1}{2} span})^2] \tag{1}$$

For trailing edge:

$$x = chord + \frac{1}{2} span\ \tan(SBA)[1 - (\frac{\bar{y}}{\frac{1}{2} span})^2] \tag{2}$$

Where$\bar{y} = \begin{cases} y\ y \leq \frac{1}{2} span \\ span - y\ y > \frac{1}{2} span \end{cases}$

Knowing that NACA0012 used as wing aerofoil section.

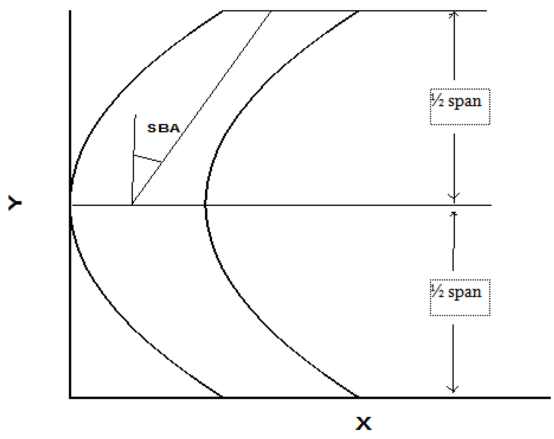


Figure 1: Wing Geometry

THEORETICAL ANALYSES

One of most used method for computing flow about arbitrary geometries are those which represent the configuration boundary with some form of source / doublet – vorticity distribution generally called panel method. This method was introduced by Hess and smith, the most significant capability offered by panel method is the treatment of complicated, multi-component geometries without any need for grid generation in the domain surrounding the configuration.

The panel method solvers considers potential flow (incompressible, irrational, invicidflow) only, so this method less accurate than other CFD solvers, but within the limiting theoretical assumptions, panel method solvers have provided results that are found to be in excellent agreement with test data and other data obtained by other CFD solvers. The most significant property of potential function is principal of super position. If each of Φ_1, Φ_2, Φ_n be an equation satisfies Laplace equation which is:

$$\nabla^2\Phi = 0 \tag{3}$$

Then the some of these equations satisfies the Laplace equation. This can be written as :

$$\boldsymbol{\Phi} = \sum_{k=1}^{n} \mathbf{c_k \Phi_k} \tag{4}$$

Where C1, C_2, C_m are arbitrary constants, and therefore:

$$\nabla^2\Phi = \sum_{k=1}^{m} c_k \nabla^2 \Phi_k = 0 \tag{5}$$

Due to this property panel method can solve problems of multi-components configuration. Shape and singularity of panels determine the order of panel method. If panel surface is not linear or singularity strength distribution not linear then it's called higher order panel method else it's called low order panel method.

Consider a body with known boundaries S_B, Submerged in a potential flow. The flow in outer region is incompressible and irrotational, so continuity equation in terms of total potential function is

$$\nabla^2\Phi^* = 0 \tag{6}$$

The general form of equation (3) can be written for a sum of sources anddoublets singularities placed on solid boundary S_B of flying object, using principle of superposition the total potential function is the sum of these singularities

effect with free stream effect.

$$\Phi^{*}(x,y,z) = -\frac{1}{4\pi}\int_{S_B}\left[\sigma\left(\frac{1}{r}\right) - \mu\, n.\nabla\left(\frac{1}{r}\right)\right] ds + \Phi_\infty \tag{7}$$

Here the normal vector n points inside the body, and Φ_∞ is the free- streampotential:

Doublets & Sources singularities used to represent non-zero thickness bodies only, while doublets only used for zero thickness bodies and so eq. (3) can be rewritten as [9].

$$\boldsymbol{\Phi}^{*} = \frac{\mathbf{1}}{\mathbf{4\pi}}\int_{body+wake}\boldsymbol{\mu\, n.\nabla}\left(\frac{1}{r}\right)\boldsymbol{ds} - \frac{\mathbf{1}}{\mathbf{4\pi}}\int_{body}\boldsymbol{\sigma}\left(\frac{1}{r}\right)\boldsymbol{ds} + \boldsymbol{\Phi}_\infty \tag{8}$$

4. BOUNDARY CONDITIONS

4.1 Far Field Condition

At very far region from the body boundary the velocity mustequal to freestream velocity i.e.

$$\mathrm{Lim}_{r\to\infty}\nabla\Phi = 0 \tag{9}$$

Where $= \sqrt{x^2+y^2+z^2}$, in potential flow this condition automatically satisfied.

4.2 Tangency Condition

This condition means that normal velocity on the body surface must equal zero, two approaches may be applied to satisfy this condition which are Neumann and Dirichlet Boundary condition.

4. 2. 1 Dirichlet Boundary Condition

This condition based on setting the total potential function everywhere on solid boundaries to be constant; consequently velocity component normal to solid boundary will be zero.This condition also named indirect condition

$$\Phi^{*}(x,y,z)|_{S_B} = constant \tag{10}$$

In case of source / doublet distribution, this condition can be satisfied by setting ($\Phi^{*}_{inner=}\Phi^{\infty}$) i.e. Inner perturbation potential equal zero.

$$\frac{1}{4\pi}\int_{body}\mu\,\frac{\partial}{\partial n}\left(\frac{1}{r}\right)ds - \frac{1}{4\pi}\int_{body}\sigma\,\left(\frac{1}{r}\right)ds = 0 \tag{11}$$

Sources strengths are determined by the following equation.

$$\sigma = \vec{n}.\vec{Q}_\infty \tag{12}$$

4.3 Kutta Condition

This condition states that circulation of flowat trailing edge equal zero. Thiscondition can be satisfied when each upper and lower velocity at trailing each is equaled and parallel.

$$\gamma_{trailing\ edge} = 0 \tag{13}$$

4.4 Panel Method Methodology

Since the integrations mentioned above cannot determined exactly, it must be determined numerically. The main steps in panel method can be summarized as follows:-

- Grid generation, which means mainly discretization on flying body surfaces to set of panels, determining collocation points (centroid) and constructing normal vectors for each panel.
- Placing quadrilateral source and quadrilateral doublet on each panel. Strength of source can be calculated from eq.(10), while doublets strength assumed to be unit.
- Constructing the influence coefficient matrix [$\mathbf{a_{ij}}$] where $\mathbf{a_{ij}}$ is the normal velocity on solid boundary induced by **j** doublet at collocation point of panel **i**.
- Constructing the RHS (right hand side) vector where RHS_i represents the sum of velocities induced by sources and free stream velocity on collocation point of panel **i**.
- Solve the algebraic equation to yields doublet's strength:

$$[\,a_{ij}\,]\begin{bmatrix} M_1 \\ \vdots \\ M_n \end{bmatrix} = \begin{bmatrix} RHS_1 \\ \vdots \\ RHS_n \end{bmatrix} \quad \ldots\ldots\ldots \quad (14)$$

- Velocity can be calculated at any point by sum of induced velocities by all sources, doublets and free stream compute.

5. RESULTS AND DISCUSSIONS

A Fortran program was constructed by authors to perform panel method steps, on curved wing. This section is devoted to present and discuss various numerical data obtained from implementation of this program. All contours were drawn with Tec plot 360 while all chart drawn with excel 2010 program.

A grid of 2160 panel was used on wing surfaces with full cosine discretization formula in both chord and span directions and make panel of 300 time chord at trailing edges panels.It should be mentioned that drag coefficient not included in this work because of neglecting viscous effect in potential flow, which is the main component of drag. While lift coefficient depends mainly on pressure distribution over the flying body, so this method can be used to estimate lift coefficient effectively.

Figures (2, 3, 4, 5, 6 & 7) show pressure coefficient contours on both upper and lower surfaces for wing with (ASR=4,SBA=30 and AOAT=5), (ASR=4,SBA=30 and AOAT=8), (ASR=6,SBA=40 and AOAT=5),(ASR=6,SBA=40 and AOAT=8),(ASR=8,SBA=50 and AOAT=10) and (ASR=8,SBA=50 and AOAT=12) respectively, it can be seen that there is symmetric behavior in pressure coefficient values about root chord axis due to geometrical symmetry. Pressure coefficient values on lower surface are greater than upper values specially near the leading edge due to stagnation point occurring in lower surface. At wing tips there is little change in pressure coefficient distribution behaviorcaused by trailing vortex formation effect.

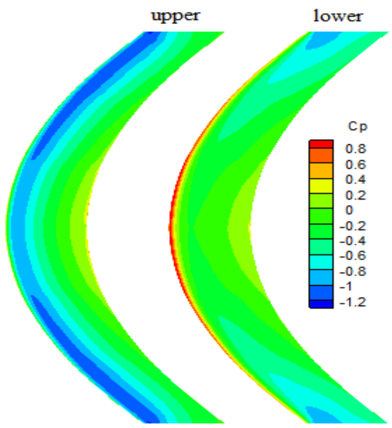
Figures (8, 9, 10, 11, 12 & 13) show local lift coefficient contours on both upper and lower surfaces for wing with with (ASR=4,SBA=30 and AOAT=5), (ASR=4,SBA=30 and AOAT=8), (ASR=6,SBA=40 and AOAT=5),(ASR=6,SBA=40 and AOAT=8),(ASR=8,SBA=50 and AOAT=10) and (ASR=4,SBA=50 and AOAT=12) respectively, the difference betweenupper and lower surfacesis due to different of normal vectors direction.

Figures (14, 15, 16, 17, 18 & 19) show velocity to free stream velocity ratio distribution contour on both upper and lower surfaces wing with with (ASR=4,SBA=30 and AOAT=5), (ASR=4,SBA=30 and AOAT=8), (ASR=6,SBA=40

and AOAT=5),(ASR=6,SBA=40 and AOAT=8),(ASR=8,SBA=50 and AOAT=10) and (ASR=8,SBA=50 and AOAT=12) respectively, also show formation of lateral velocity component directed away from root chord axis due to sweeping back angle.

Figure (20, 21, 22, 23 & 24) show that lift coefficient increases with angle of attack because of symmetric profile of wing used (NACA 0012), also its obvious that when sweep back angle maintained constant the lift coefficient increases with increase in aspect ratio for same angle of attack because the decreases in ratio of area that endures trailing vortex at wing tips to total area of wing. As sweep back angle increases the difference in lift coefficient valuesat different aspect ratios decreases this may be due to of increase in lateral velocity induced values.

Figures (25, 26, 27 & 28) show the decrease of lift coefficient as sweep back angle increases this due to increases in lateral velocity formation on wing.



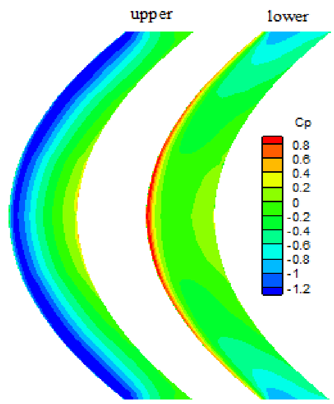
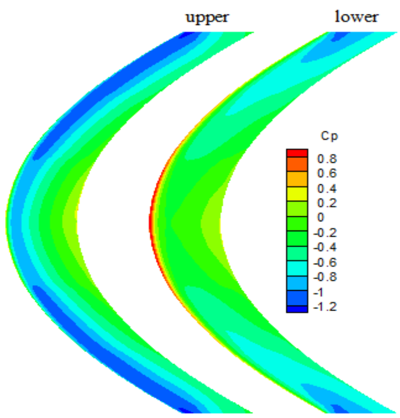


Figure 2 **Figure 3**



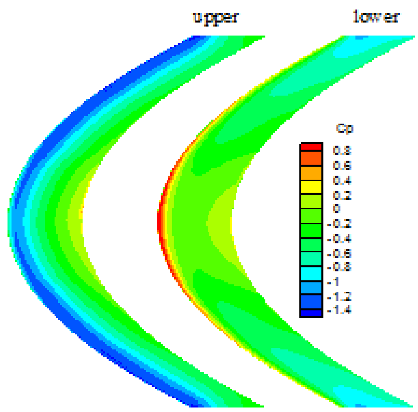
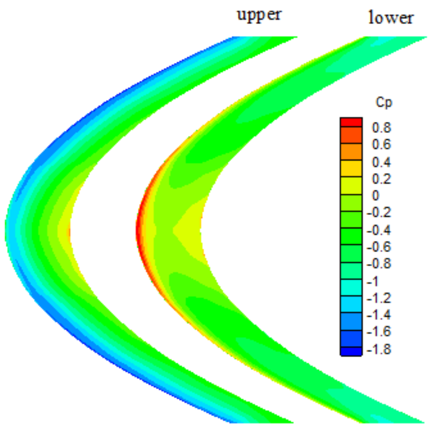


Figure 4 **Figure 5**



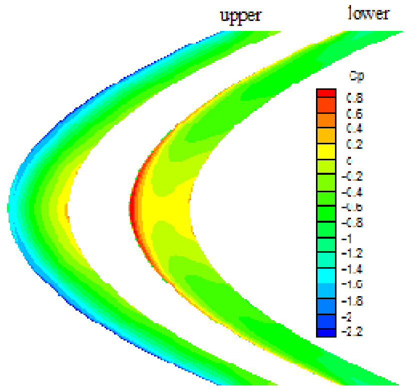
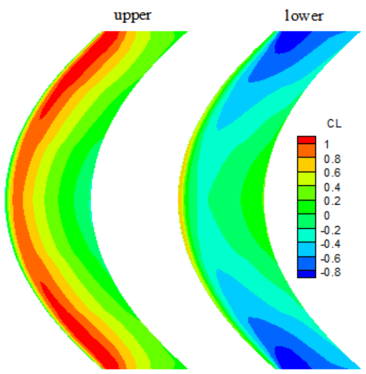


Figure 6 Figure 7



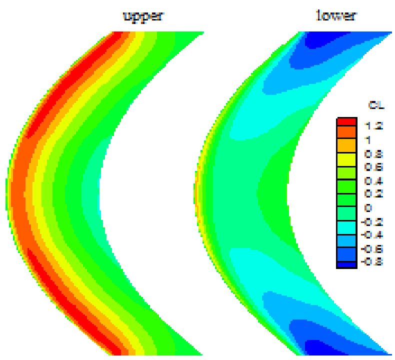
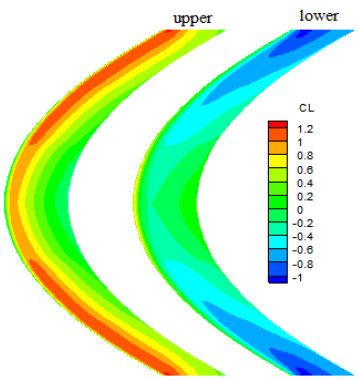


Figure 8 Figure 9



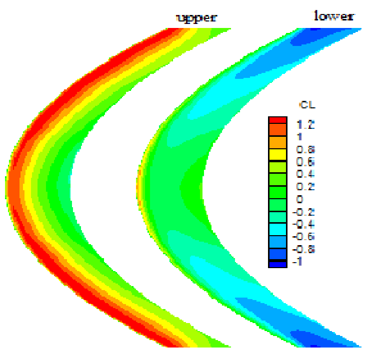
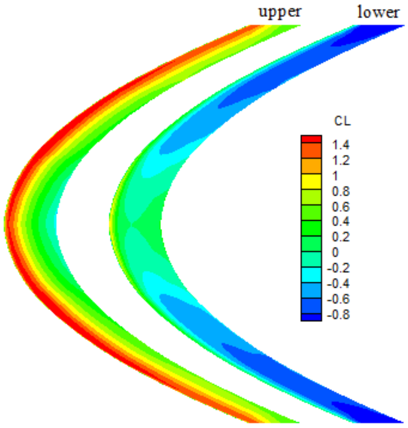


Figure 10 Figure 11



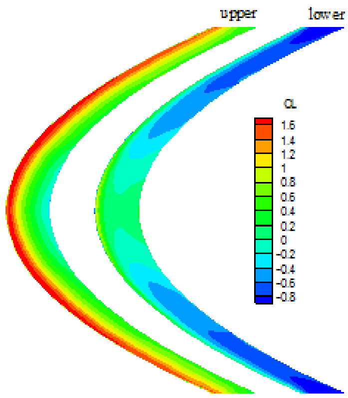
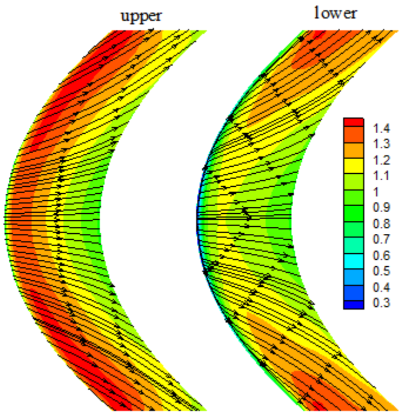


Figure 12 Figure 13



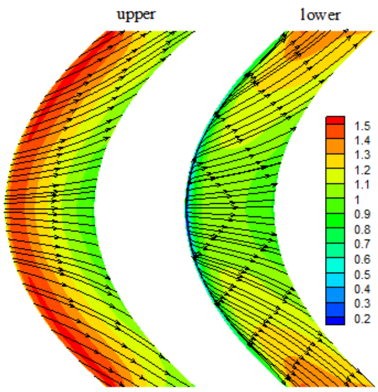
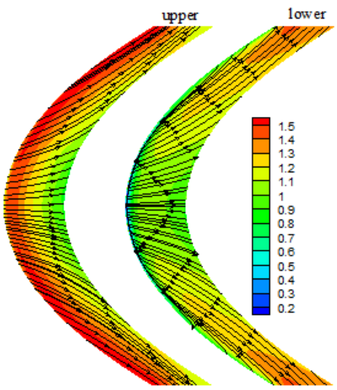
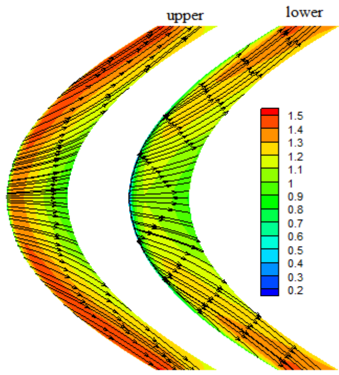


Figure 14 Figure 15





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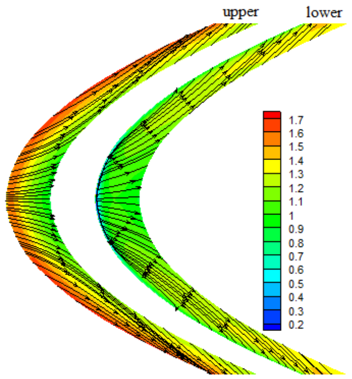

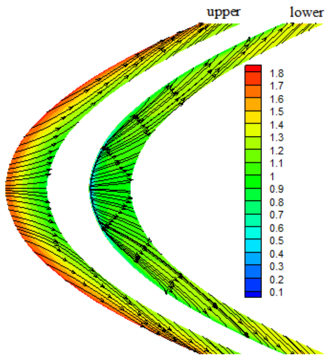

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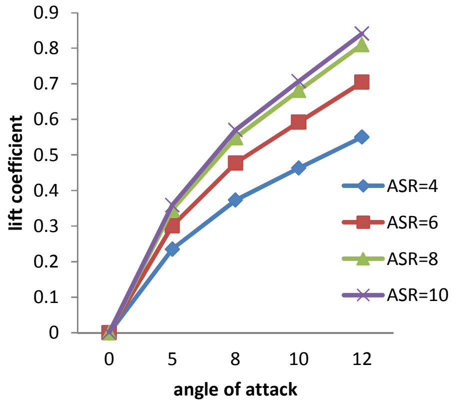

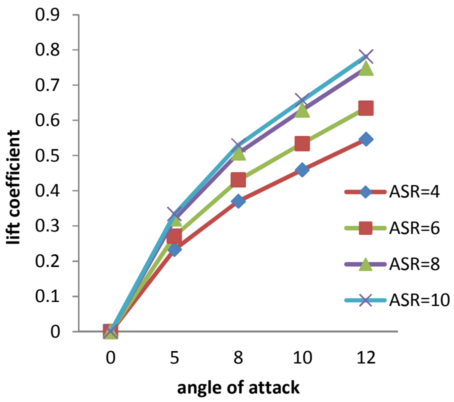

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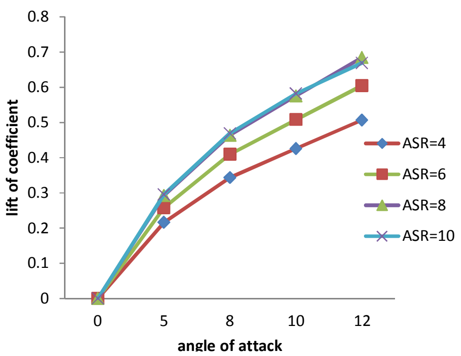

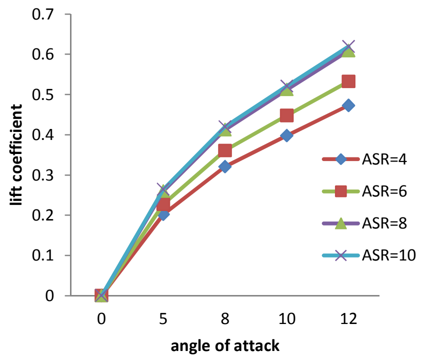

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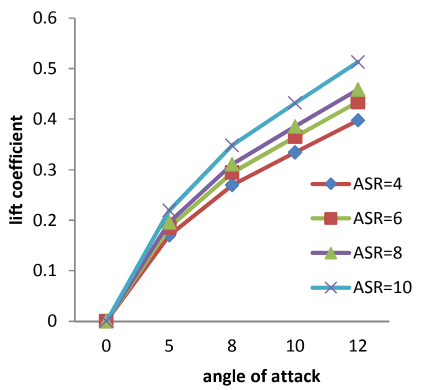


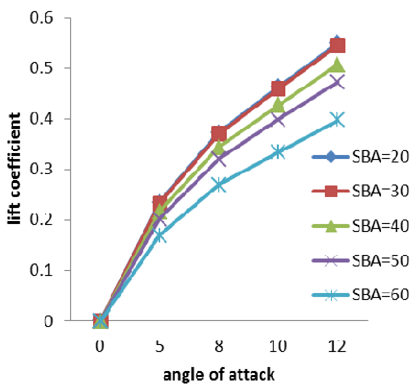
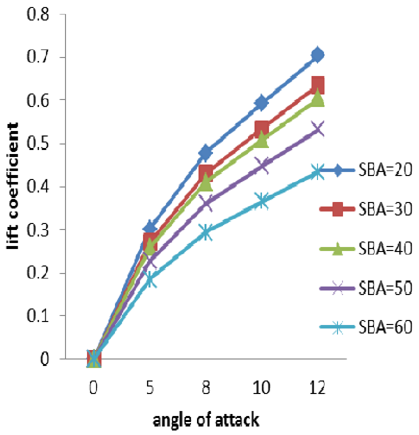


Figure 24 Figure 25



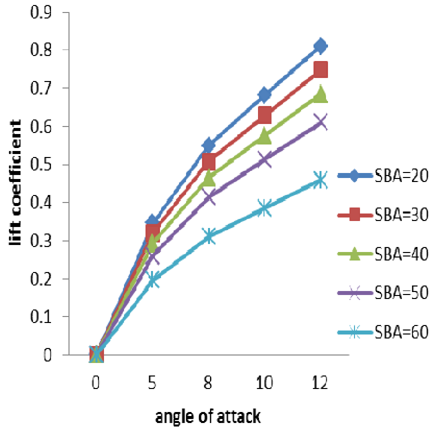


Figure 26 Figure 27

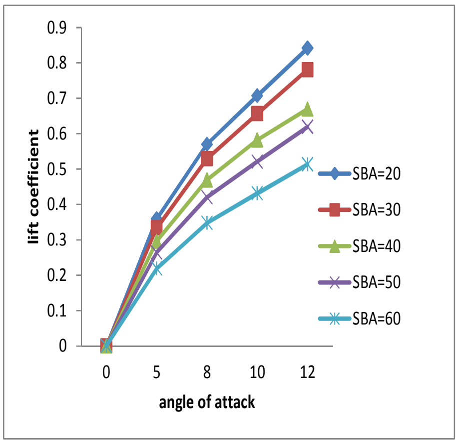


Figure 28

5. CONCLUSIONS

This work presents the simulated flow about curved wing and it was observed that the lift coefficient increases as aspect ratio increases because of reduction ratio of area effected with trialing vortex to total wing area. Lift coefficient decreases as sweep back angle increases due increase in lateral velocity induced directed away from root chord axis.

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